Small Launch Vehicle Design Optimization

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Abstract

Currently, small satellites (e.g. nanosatellites) can be included as secondary payloads in large launch vehicles housing much larger primary payloads. In an interest to minimize the complexity associated with having multiple payloads and to increase the chance of successful payload deployment into orbit; the use of smaller launch vehicles specifically designed for these small satellites is preferential. Pursuant to minimizing the cost per mass of payload to low earth orbit, it is desired that the launch vehicle mass be minimal (optimized) at liftoff. Options to consider in the design of the launch vehicles include the type of propulsion systems utilized, the number of stages used, and the materials selection in consideration to vehicle loads which will affect the inert mass fraction. A code was developed that used a modified form of the ideal rocket equation to calculate the gross liftoff mass of a two-stage launch vehicle. Parameters such as payload mass, total velocity required to reach orbit, specific impulse of each stage, and the propellant mass fraction of each stage are input into the code. A plot showing the initial vehicle mass versus the fraction of the total velocity produced by the first stage is then generated. The minimum mass obtainable with the given configuration is also displayed in the output. This code only provides an initial estimate of the vehicle mass since it uses the ideal rocket equation. Gravity losses, steering losses, and drag losses are not included in the initial equation; to mitigate these concerns, a higher total velocity (30,000 ft/s or 9,144 m/s) is initially assumed in an effort to prevent an underestimation of launch vehicle size. Future work would include expanding the code to consider the above velocity losses in detail and for launch vehicles having more than two stages in order to produce results with higher fidelity.

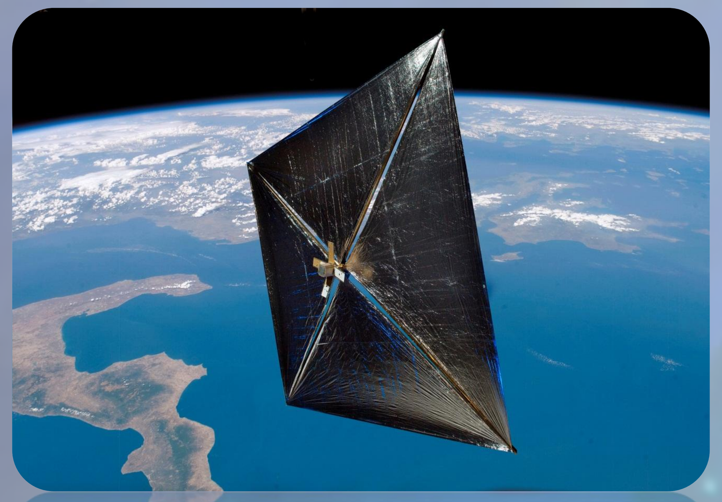

Figure 1. Artists Concept of a Solar Sail in Space; an Example of a Nanosatellite

Methodology

The analysis began with the ideal rocket equation

$$\Delta v = -v_e \ln\left(\frac{m_f}{m_i}\right) \qquad (1)$$

Where

- v_e = exhaust velocity of propellant mass
- m_f = final vehicle mass
- m_i = initial vehicle mass
- Δv = change in velocity

The ideal rocket equation was then manipulated to find the mass of the propellant as a function of several parameters.

$$m_{prop} = \frac{m_{pay}\left(e^{\left(\frac{\Delta V}{I_{sp} g_0}\right)} - 1\right)(f_{prop})}{1 - (1 - f_{prop})e^{\left(\frac{\Delta V}{I_{sp} g_0}\right)}} \qquad (2)$$

Where

- I_{sp} = specific impulse
- m_{prop} = propellant mass
- f_{prop} = propellant mass fraction
- m_{pay} = payload mass
- g_0 = acceleration due to gravity

An interactive code was then constructed in MATLAB for a two stage launch vehicle that requested inputs from the user. Parameters requested include: payload mass, total delta V needed, I_{sp} of each stage, propellant mass fraction of each stage, the minimum and maximum delta V fractions needed from the first stage, and the desired resolution of the results (i.e. number of data points). Calculations were iterated over the given range of delta V and the initial vehicle mass was determined at each location. A plot showing the initial vehicle mass as a function of the delta V fraction of the first stage is displayed as well as the minimum value of the initial vehicle mass.

Results and Discussion

Several different cases were analyzed with the code; configurations included solid/liquid staging, and liquid/liquid staging. Below, Table 1 shows some of the relevant properties for various propellants that were utilized and Table 2 shows the cases that were originally analyzed.

Table 1. Properties of Selected Propellants

Propellant Type	Engine Type	Specific Impulse (I_{sp}) (s)	Mixture Ratio	Propellant Mass Fraction
RP-1/LO_2	Gas Generator	300	1.6	0.90
LH_2/LO_2	Gas Generator	410	6.2	0.88
Hydrazine/ N_2O_4	Pressure Fed	285	4.0	0.85
HTPB/AL/ AP	Solid	265	--	0.90

Table 2. Cases Analyzed with the Code

	Stage 1		Stage 2	
Case	Fuel	Oxidizer	Fuel	Oxidizer
1	RP-1	LO_2	RP-1	LO_2
2	LH_2	LO_2	LH_2	LO_2
3	LH_2	LO_2	Hydrazine	N_2O_4
4	Aluminum/ HTPB	Ammonium Perchlorate	RP-1	LO_2
5	Aluminum/ HTPB	Ammonium Perchlorate	LH_2	LO_2

Results were plotted for each of the several cases. In Figure 2 below, you can see a sample plot generated by each of the cases using a total delta V of 9,144 m/s and a payload of 1 kg.

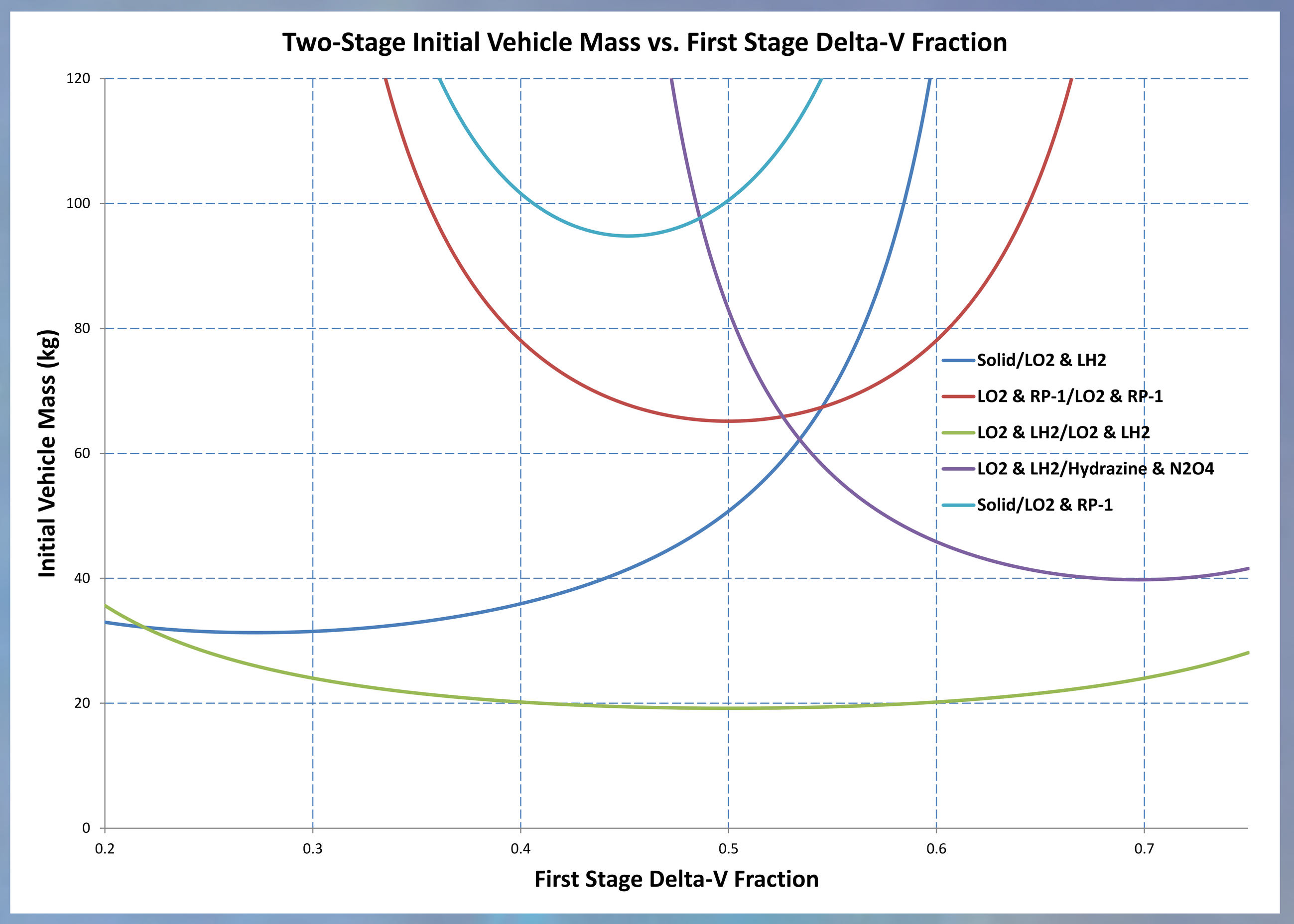

Figure 2. Plot of Initial Vehicle Mass vs. First-Stage Delta V Fraction

Table 3. Minimum Initial Vehicle Masses

Propellant Type	Minimum Initial Mass (kg)
RP-1 & LO_2/RP-1 & LO_2	65.150
LH_2 & LO_2/LH_2 & LO_2	19.193
LH_2 & LO_2/Hydrazine & N_2O_4	39.770
Aluminum & Ammonium Perchlorate/RP-1 & LO_2	94.800
Aluminum & Ammonium Perchlorate/LH_2 & LO_2	31.298

Given the results of the vehicle masses and knowing other parameters of the vehicle such as I_{sp}, further analysis can be done. Preliminary thrust profiles can be designed to meet vehicle requirements and then these profiles can be analyzed with a trajectory analysis code that can determine if the design of the flight profile is sufficient to reach orbit.

Future Work

- Expand code capabilities to include n-stages, not just two
- Enable the code to automate calculation for a range of I_{sp} and f_{prop} values for each stage
- Account for losses associated with gravity, steering, and drag in greater detail:

$$v_{LEO} = \underbrace{\int_{ign}^{bo} \frac{F}{m} dt}_{(\Delta v_{prop})} - \underbrace{\int_{ign}^{bo} \frac{F}{m}(1 - cos\alpha')dt}_{(\Delta v_{steering})} - \underbrace{\int_{ign}^{bo} \frac{D}{m} dt}_{(\Delta v_{drag})} - \underbrace{\int_{ign}^{bo} g_l sin\gamma dt}_{(\Delta v_{gravity})}$$

References

[1] Sutton, George P., and Oscar Biblarz. *Rocket Propulsion Elements*. New York: John Wiley & Sons, 2001.

[2] Henry, Gary N., Wiley J. Larson, and Ronald W. Humble. *Space Propulsion Analysis and Design*. New York: McGraw-Hill, 1995.